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THERMAL BARRIER COATINGS - A NEAR TERM, HIGH PAYOFF TECHNOLOGY

by S. R. Levine and J. S. Clark
Lewis Research Center and
U.S. Army Air Mobility R&D Laboratory
Cleveland, Ohio 44135

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Stanley R. Levine and John S. Clark

Lewis Research Center and U.S. Army Air Mobility R&D Laboratory

Cleveland, Ohio

ABSTRACT

A duplex plasma sprayed thermal barrier coating consisting of an inner 5 mil NiCrAlY bond coat and an outer 15 mil stabilized zirconia ceramic was successfully tested in a J-75 research engine at the NASA-Lewis Research Center. In this test, the coating was applied to the first stage turbine rotor blades and was cycled for 500 two-minute cycles at coating surface temperatures of 1065°C in a clean fuel environment. Addition of the thermal barrier coating to current industrial gas turbine coating configurations can result in airfoil metal temperature reductions up to 110°C. The coating also has potential for resisting corrosion caused by impurities found in heavy fuels. Studies indicate that turbine inlet temperatures may also be increased with thermal barrier coatings with no increase in cooling air requirements or substrate metal temperatures. These factors plus potential retrofit and short development time make thermal barrier coatings extremely attractive for a near term heavy fuel fired utility gas turbine. The proposed development plan for this technology includes determining coating durability in heavy fuels with various levels of clean-up, developing coating application technology, building a sufficient engineering data base to design with confidence and demonstrating the durability of the coating in utility service.

INTRODUCTION

Thermal barrier coatings (TBC) were investigated at the Lewis Research Center for protection of critical hot section components of gas turbines in the early 1950's (1,2). They have also been investigated for protection of cooled rocket nozzles (3). Recent investigations at Lewis (4) and elsewhere (5-7) have focused on stabilized zirconia based graded and duplex thermal barrier coatings for protection of gas turbine components. Recently, Liebert and Stecura (8) of Lewis developed and successfully engine tested a plasma sprayed duplex thermal barrier coating system. This simple two-layer coating system consists of a 0.010 cm (4 mil) NiCrAlY bond coat and a .025 to .075 cm (10 to 30 mil) 12 w/o yttria stabilized zirconia overlayer. The concept is depicted in figure 1. The stabilized zirconia ceramics have a thermal conductivity approximately 3 percent that of gas turbine alloys (4) and thus can

impose an effective thermal barrier between high temperature combustion gases and cooled turbine components--combustor liners, turbine blades and vanes. Substantial metal temperature reductions can be achieved by the addition of a thermal barrier coating. Large benefits can thus be realized in the form of higher turbine inlet temperature, reduced cooling air, longer life or simplification of the cooling scheme.

The thermal expansion coefficient of stabilized zirconia ceramic matches that of gas turbine alloys better than most other ceramics; this undoubtedly contributes to its adherence at high temperature. The NASA duplex coating uses an oxidation/hot corrosion resistant Ni-16 w/o Cr-5.6 w/o Al-0.6 w/o Y bond coat rather than the Ni-Al solid solution, Ni-20Cr or superalloy bond coats used heretofore. Further improvements in both hot corrosion and oxidation resistance may be obtained by going to other NiCrAlY compositions or switching to CoCrAlYs and increasing bond coat density.

The purpose of this paper is to review the present status of thermal barrier coatings. This review will include both experimental results and forecasts of the benefits that may be derived from use of these coatings in aircraft and utility gas turbines. Emphasis will be placed on utility gas turbines. Also, the potential of thermal barrier coatings relative to structural ceramics will be discussed. Finally, a development plan for these coatings will be outlined.

CURRENT STATUS

In this section, the results of laboratory rig and engine tests conducted at NASA-Lewis and elsewhere will be reviewed.

Coating Development

Coating development at NASA-Lewis is being carried out with the aid of furnace screening tests on solid coupon specimens. The better coatings are then tested in torches and Mach 0.3 and Mach 1 burner rigs.

The results of cyclic furnace screening of various oxide ceramic thermal barrier layers are presented in Table I (9). In this test, the specimens were heated to 975°C in four minutes, held at temperature for one hour and cooled to 280°C in one hour. Of the oxides tested, yttria-stabilized zirconia is clearly the best. This oxide was not as widely studied in the thermal barrier application (4-7) as the other oxides listed in Table I due to its higher cost and lower availability. In figure 2, photomicrographs of the coating in the as-deposited condition and after the cyclic furnace test are shown (9). Since this coating was applied manually, there is about 0.005 cm (2 mils) of variation in thickness of the bond coat and oxide from specimen to specimen and location to location on a specimen. This variation is evident with the bond coat shown in figure 2(a). Another aspect is the porosity of the bond coat and oxide as-deposited. After the furnace exposure, it appears that some sintering of the ceramic occurred even at the relatively low temperature of this test. There was also some oxidation of the NiCrAlY bond coat.

In Mach 0.3 burner rig tests of cooled J-75 turbine blades, the ranking of the

oxides as to performance was the same as in the cyclic furnace tests. In the Mach 0.3 burner, the J-75 blades were heated nonuniformly and very severe local surface temperature gradients developed. The yttria stabilized zirconia coating survived as many as 3200 cycles consisting of 80 seconds at 1280°C surface temperature and a 915°C substrate temperature followed by cooling to 75°C. The yttria stabilized zirconia coating also survived 246 cycles consisting of one hour at a 1410°C surface temperature and 900°C substrate temperature followed by cooling to 75°C. Both tests were terminated due to erosion of the coating to about half its initial thickness (9). This erosion, attributed to carbon from the Jet A fueled burner, is of considerable concern. In tests carried out in a natural gas fired Mach 1 burner rig erosion of the coating was not evident.

Engine Test Results

As illustrated in figure 3(4), metal temperature reductions of about 200°C were achieved at the leading edge of vanes in a J-75 research engine with a 0.028 cm (11 mil) thick zirconia-calcia coating applied. The coating over a nichrome bond coat survived 150 hours with 35 start-stop cycles and gas temperatures as high as 1370°C. Steady state wall temperatures were as high as 650°C with transients to 925°C (4). Inlet pressure was 3 atm. This study by Liebert and Stepka also indicated that the effectiveness of thermal barrier coatings increases as the turbine inlet pressure increases. For example, they calculated that for a 0.020 cm (8 mil) thick zirconia layer on a vane with metal temperature maintained at 990°C, the potential turbine inlet temperature increase in a 40 atm. core engine was 560°C, whereas in the 3 atm research engine the increase is only 80°C.

The success of the vane test provided the impetus to test the improved NiCrAlY bond coat in conjunction with calcia, magnesia, and yttria stabilized zirconia thermal barriers on J-75 turbine blades in the research engine. After 500 two-minute cycles between full power and engine flameout, all coated blades were in good condition, as can be seen from frigure 4(10). At full power, engine conditions were 1370°C turbine inlet temperature (TIT), 3 atm and 8300 rpm and at flameout the conditions were 730°C, 1 atm and 3300 rpm. At full power, the coating surface temperature was as high as 1080°C, the blade metal temperature was as high as 930°C and the temperature drop through the coating was as high as 135°C. At flameout, the blade metal temperature was 530°C.

Because of the roughness of the as-plasma spray deposited thermal barrier coating significant aerodynamic losses occur. These losses can be reduced almost to the same level as those encountered with a blade without the TBC by a simple polishing procedure (11).

Other Test Results

JT8D combustor liner tests. Recently, tests have been run at NASA-Lewis on a JT8D combustor liner, with and without the thermal barrier coating (11). Combustor pressure was about 17 atmospheres, inlet temperature was 440°C and exit temperature was maintained at about 1065°C for both cases. Two fuels were used, Jet A and blended Jet A to include about 60 percent aromatics. In the Jet A test, maximum liner metal

temperatures were reduced by about 165°C by the thermal barrier coating. For the blended Jet A plus aromatics maximum metal temperatures were reduced by about 210°C. Also, for certain operating conditions, visible smoke was decreased slightly as was flame radiation. This was believed to be a result of the higher reflectivity of the TBC (0.8 compared to 0.2 for metallic liners) resulting in more complete combustion of carbon particles. There was no measurable reduction in HC, CO, or NO_x for these operating conditions.

Hot corrosion behavior. At Lewis, IN-713C was coated with ZrO_2.5 w/o CaO by plasma spray deposition. Bare and zirconia coated specimens were exposed to 900°C cyclic furnace hot corrosion induced by coating the specimens with 1 mg/cm^2 of Na_2SO_4. After 100 one-hour cycles, the zirconia coated specimen gained 2.1 mg/cm^2 whereas the bare specimen lost 73.8 mg/cm^2. Figure 5 shows the results of a 900°C, M 0.3 burner rig hot corrosion test of the NASA duplex TBC and plasma sprayed NiCrAlY bond coat protected René 41 sheet specimens. The fuel was Jet A with 5 ppm of sea salt injected in the burner. Since the sheet thickness was 0.15 cm and the specimen was uncooled, anticipated difficulties with zirconia adherence were encountered. After 40 cycles cracking and some spalling was evident at the corners of the specimen which were out of the hot zone. These defects propagated down the edges and after 70 cycles, the coating started to spall from the major surfaces in the hot zone. After 300 hours of exposure in this accelerated test, the visual appearance of the coated specimens was good, whereas reduction in thickness of the bare René 41 specimen was apparent after 100 hours. Microstructurally, after 300 hours, the NiCrAlY layers were nearly totally consumed and hot corrosion attack had penetrated to a depth of 0.01 cm/side. In a similar test of 200 hours duration, dense oxide dispersion strengthened NiCrAl of composition similar to the bond coat lost only about 1 mg/cm^2 (13).

In a study by Dapkunas and Clarke (5) at the Naval Ship R&D Center, zirconia stabilized with either 3 w/o CaO, 3 w/o MgO, or 8 w/o Y_2O_3 was exposed at 900°C in molten Na_2SO_4 for times up to 1000 hours with only minor attack. In another phase of this program, cooled graded thermal barrier coated specimens with a NiAl metallic phase were given two 25-hour exposures in a burner rig. The surface temperature was 1040°C and the metal temperature was 815°C. The fuel was diesel (1 percent sulfur) and 100 ppm of sea salt was injected. Little sulfidation attack of the metallic or ceramic phase on the MarM 509 substrate occurred. The authors (5) attributed this to the high surface temperature of the specimen retarding sulfidation of the cooler interior.

Based on these results, the thermal barrier coatings hold significant promise for operation in dirty fuel and/or air environments. Additional investigations of the tolerance of thermal barrier coated materials to corrosion by fuel and air impurities are being conducted at NASA-Lewis. Additional dirty fuel tolerance studies are being carried out at the Department of the Navy and at Westinghouse in cooperation with NASA.

Erosion behavior. In a study of graded thermal barrier coatings, Cavanagh et al. of Detroit Diesel Allison Division (6) point out that the erosion resistance of zirconia is a function of density with a 100 percent dense material being about six times better than a plasma sprayed deposit. However, the dense material was not as tolerant to ballistic impact. Thus, there appears to be a density tradeoff between erosion resitance and the ability of a porous coating to absorb impact energy and arrest crack propagation.

Utility Gas Turbine Studies

NASA studies. The effects on performance of adding a 0.038 cm (15 mil) TBC to a recuperated gas turbine system with a turbine inlet temperature of 1200°C, recuperator effectiveness 0.85, relative recuperator pressure drop, $\Delta P/P$, 0.03, and turbine and compressor polytropic efficiency of 0.9, and a pressure ratio of 10.0 (base case figure 6). The TBC reduced the cooling flow requirements substantially which resulted in an efficiency improvement of 3.2 percent and a specific power improvement of about three percent relative to the base case. Similar calculations were performed for the same machine with ceramic vanes and blades. The results are also shown in Figure 6. With monolithic ceramic vanes and air cooled blades, a four percent relative improvement in efficiency and a three percent relative improvement in specific power result. Similarly, the same engine with ceramic blades and ceramic vanes would give a 6.9 percent improvement in efficiency and a 6.7 percent improvement in specific power compared to the base case

Also shown on figure 6 is the effect of derating (TIT reduction) currently required to burn heavy fuels. It was assumed that a TIT reduction from 1200°C to 1040°C is necessary for this machine when burning heavy fuel. For this case, the efficiency decreased by 3.2 percent and the power output decreased by 16.7 percent, relative to the base case.

Westinghouse TBC study. A study was made at Westinghouse - Generation Systems Division funded by NASA to evaluate, by means of cycle efficiency and cost-of-electricity calculations, the effect of the TBC on Westinghouse current production and proposed advanced design gas turbine systems. Westinghouse evaluated simple cycle peaking gas turbines, recuperative cycle gas turbines and combined cycles (14). The performance calculations for combined cycles are summarized in figure 7. The current production W-501-D gas turbine was used as a base for all of the calculations. Retrofit of the current W-501-D configuration with a 0.038 cm (15 mil) TBC was calculated to result in a 13.3 percent reduction in cooling air flow (not shown in figure), and 0.57 percent improvement in heat rate, and a 1.1 percent increase in specific power output, all relative to the base case. Similarly, with the same cooling configuration and flow rate, it was found that the turbine inlet temperature could be uprated to 1205°C, resulting in about a three percent improvement in heat rate and a six percent increase in specific power. To uprate to even higher turbine inlet temperatures, a more advanced impingement, convection air-cooling configuration was required. This advanced configuration permitted a turbine inlet temperature of 1315°C, an improvement of 5.4 percent in heat rate, and about 12 percent increase in specific power, in spite of a 45 percent increase in cooling air. Similar, but smaller, benefits were obtained for recuperated cycle systems and simple cycle machines.

To put these performance improvements in perspective, a 300 MW combined cycle plant, operating at 65 percent capacity factor and burning distillate fuel, would save about $450,000 per year in fuel costs for each percent improvement in heat rate.

Cost-of-electricity (COE) results are shown in figure 8 for various turbine inlet temperatures for the combined cycle. In each case, the addition of the TBC reduced the COE and increases in turbine inlet temperature resulted in further reduction in COE. With the 1315°C T_{IT}, a 5.6 percent reduction in COE relative to the base case was obtained for the combined cycle case.

Westinghouse also calculated the effect on COE of using a residual fuel in a current temperature machine with the TBC. For the combined cycle case a 7.4 percent reduction in COE resulted from the lower fuel cost, which is a greater improvement than increasing turbine inlet temperature to 1315°C (5.6 percent). Thus, one of the most significant potential benefits of the TBC might be to allow operation of current turbines with residual fuels

United Technologies Corporation (UTC) TBC study. A study at UTC is underway to investigate the potential of thermal barrier coatings on the FT-50 gas turbine engine (15). ERDA/Division of Conservation Research and Technology is funding this effort through an Interagency Agreement with NASA. The FT-50 gas turbine engine is an advanced high-temperature machine. Two prototypes have been built. This advanced gas turbine is a high-pressure (16:1 pressure ratio), three-spool machine with a free power turbine. The baseline case for the study was the FT-50 A4 configuration with maximum hot-spot metal temperatures of 870°C, a higher heating value heat rate of 10200 J/w-hr (thermal efficiency of 35.4 percent) and total cooling air flow equal to 21.8 percent of compressor inlet flow rate.

For the same maximum hot spot metal temperature, 870°C application of 0.038 cm (15 mils) of the TBC allowed a reduction in total cooling air flow of 37 percent. This cooling air flow reduction resulted in an improvement in thermodynamic efficiency of 0.5 percent and an improvement in specific power output of 4.2 percent for the simple cycle while simplifying the cooling configuration. Leading edge film-cooling holes (showerhead configuration) were replaced by internal impingement and convection cooling. This change will eliminate potential leading edge film-cooling hole-plugging problems that could result from burning heavy fuels or fuels requiring the use of magnesium inhibitors to prevent vanadium induced corrosion.

Aircraft Gas Turbine Benefits

The benefits associated with thermal barrier coatings in aircraft gas turbines are being addressed by studies being carried out under the Energy Conservative Engines Office at Lewis. The coatings may be retrofitted into existing engines or into a new energy efficient engine. Preliminary studies indicate that if the coating is utilized in a manner such that turbine inlet temperature is increased 80°C while cooling air flow is reduced 40 percent, the benefits with current blade and vane alloys are an 18 percent increase in thrust, a four-fold increase in part life, and a two percent decrease in fuel consumption. For a 300 aircraft fleet savings of about $25 M/year could be realized.

DEVELOPMENT PLAN FOR UTILITY GAS TURBINES

The potential near term performance benefits offered by thermal barrier coatings in terms of dirty fuel tolerance, improved durability and reduced cooling air combined with their ability to be retrofitted into existing machines and a potentially short development time have provided a strong incentive to proceed with development. A development plan is proposed. The goals of this plan are to achieve production readiness of the coating by the beginning of 1981. This will include the verification of the coatings' capability by tests on site at a utility. The coating will also be ready, at this time, for incorporation in heavy fuel fired utility gas turbine designs as determined by the manufacturers of such machinery.

The proposed development logic is outlined in figure 9. Inputs to the thermal barrier coating development program will come from continuing benefits studies, the needs of component designers and the characteristics of potential fuels. The resistance of thermal barrier coated turbine alloys to environments ranging from fired distillate to fired dirty fuels will be examined. The effects of temperature, pressure, contaminant levels and coating variables will be determined. The response of the coating to utility gas turbine cleaning procedures will also be examined. Processes for economic production, repair and refurbishment, and NDE of the coating will be developed. NASA is currently supporting a program to develop a computer numerically controlled five axis blade coater under Contract NAS3-20112. This will provide a method for uniformly and reproducibly applying plasma spray coatings including the duplex or graded thermal barrier coatings. The engineering data base task will provide the coating physical and mechanical property information required for component design.

Successful performance of the coating in the fuel tolerance tests will warrant proceeding to verification of coating performance, first with distillate, in a manufacturer's test bed engine. A 2500-hour endurance test with distillate will then be run on-line at a utility site. The same procedure will be used to carry out a heavy fuel test. In these tests, the coating will be applied as an add-on to partial sets of blades and vanes.

CONCLUDING REMARKS

Thermal barrier coatings offer a near term, high payoff technology for application to current and near-term gas turbines. They have the potential to reduce corrosion caused by fuel impurities, improve durability, and reduce heat rate. With Government and industry cooperation, development could be completed in the early 1980's. Thermal barrier coatings can be viewed as an intermediate step on the road to structural ceramic components. The latter offer a higher payoff, but are longer term.

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TABLE I. - CYCLIC FURNACE EVALUATION OF VARIOUS ZIRCONIA THERMAL BARRIER COATINGS ON Ni-16Cr-6Al-0.6Y BOND COATING TO 975° C (REF. 9)

Alloy	Cycles to failure[a] - First visible crack, spall, etc.			
	ZrO_2-12Y_2O_3	ZrO_2-3.4MgO	ZrO_2-5.4CaO-P[b]	ZrO_2-5.4CaO-T[c]
DS MAR-M-200 + Hf	[d]673	460	255	78
MAR-M-200 + Hf	[d]650	450	255	87
MAR-M-509	[d]558	450	196	76
B-1900 + Hf	[d]628	438	226	--

[a]Cycle, 1-hr at temperature and 1-hr to cool to 280° C.

[b]P, partially stabilized zirconia derived from ZrO_2 and $CaCO_3$ spray powders (cubic and monoclinic phases).

[c]T, totally stabilized zirconia derived from stabilized spray powder (cubic phase).

[d]No failure observed.

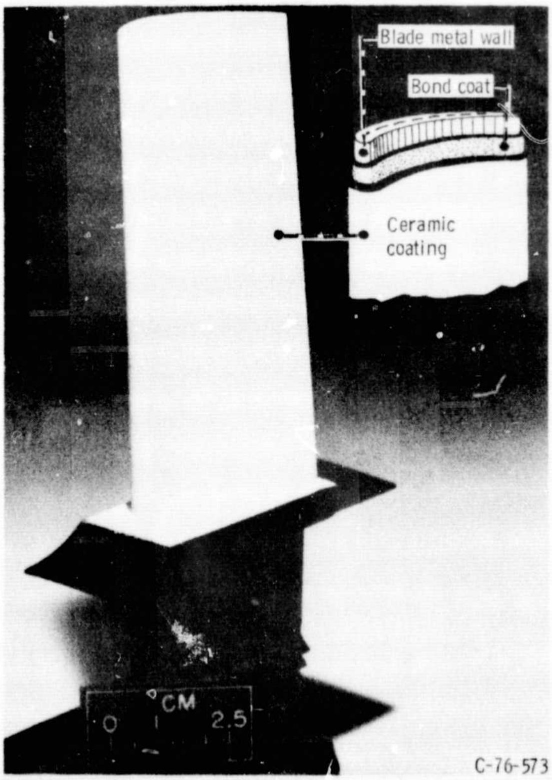
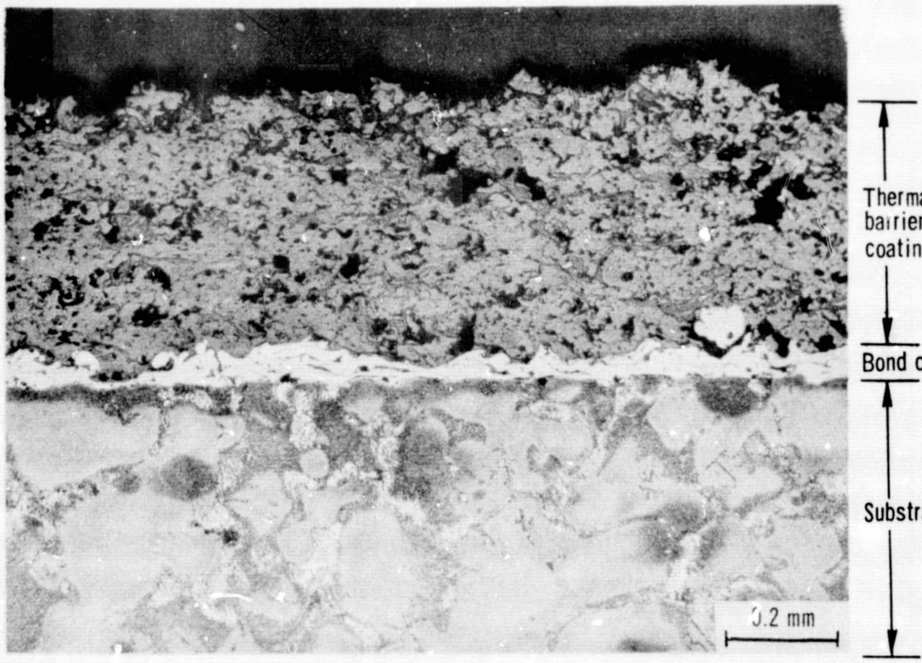
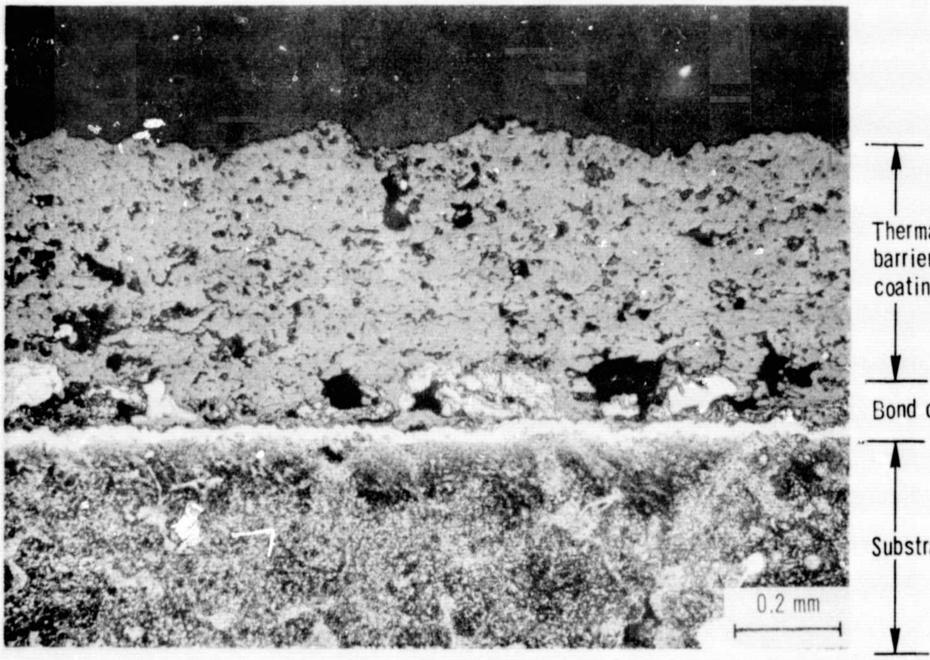


Figure 1. - Ceramic coated turbine blade.



(a) Before testing.



(b) After testing for 673 cycles at 975° C. (Cycle, 4 min heat up, 60 min at temperature, and 60 min cooling.

Figure 2. - Light optical photomicrograph of DS MAR-M-200 with Hf coupon specimen coated with Ni-16Cr-6Al-0.6Y bond coating and ZrO_2-$12Y_2O_3$ thermal barrier coating (9).

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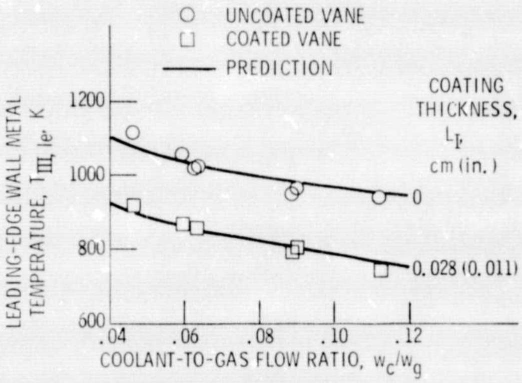

Figure 3. - Comparison of calculated and measured midspan leading-edge wall metal temperatures of uncoated and zirconia-coated turbine vanes operating in a research engine. Inlet gas temperature, 1644 K (2500° F); inlet gas pressure, 3 atm; coolant temperature, 319 K (114° F) (4).

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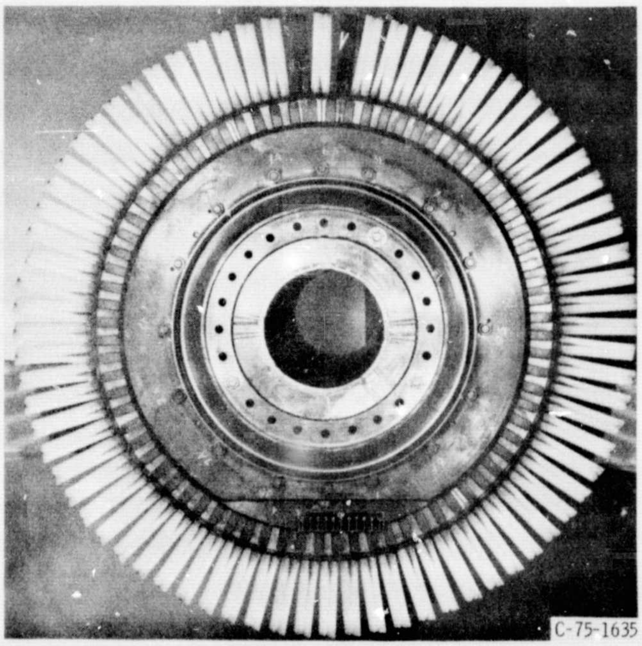

Figure 4. - Ceramic coated turbine blades after 500 cycles of testing (10).

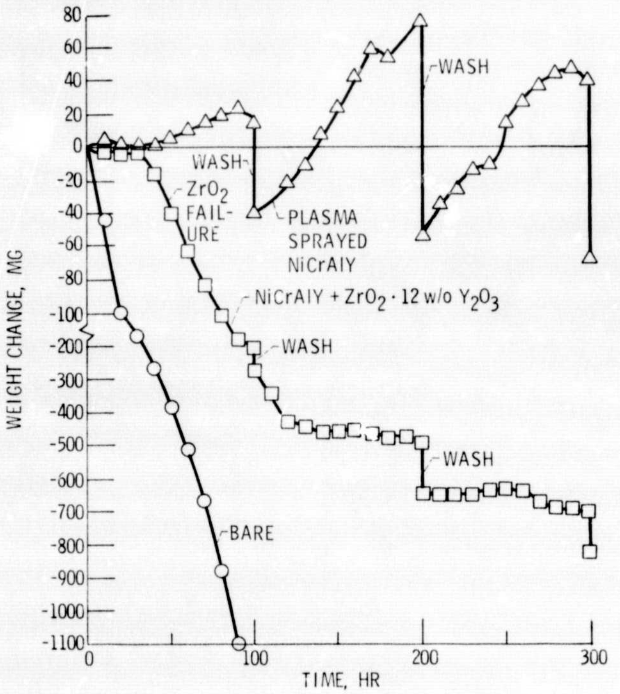

Figure 5. - Mach 0.3 burner rig hot corrosion of duplex thermal barrier and NiCrAlY bond coat protected Rene 41. 900° C, 1-hour cycles, jet A + 5 ppm sea salt. Water wash after each 100 hours. Hot zone = 6 cm^2. (S. Levine, unpublished data.)

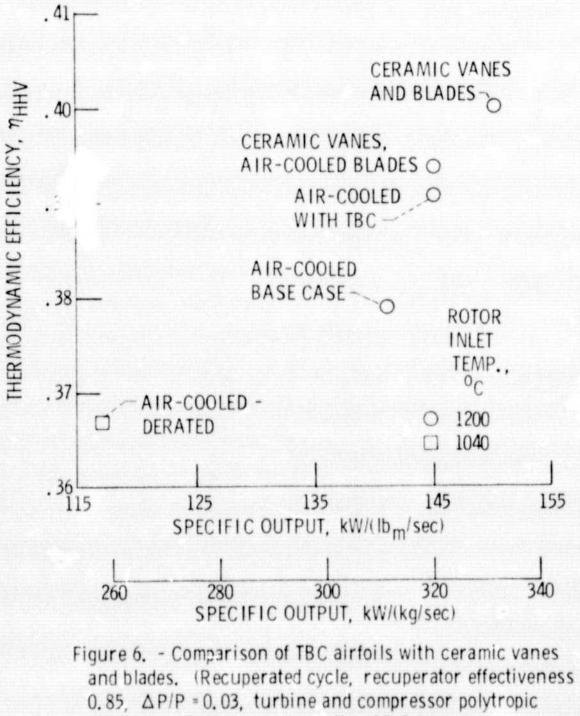

Figure 6. - Comparison of TBC airfoils with ceramic vanes and blades. (Recuperated cycle, recuperator effectiveness 0.85, $\Delta P/P = 0.03$, turbine and compressor polytropic efficiency = 0.90, pressure ratio = 10.0.)

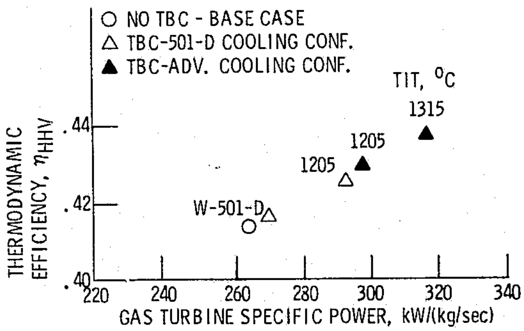

Figure 7. - Effect of TBC on performance of the W-501-D combined cycle (14).

	ENGINE CONFIGURATION	COE, mills/kW-hr
COMBINED CYCLE (0.65 CAPACITY FACTOR)	CURRENT W-501-D WITHOUT TBC - BASE CASE	33.7
	CURRENT W-501-D WITH TBC	33.6
	W-501-D UPRATED TO 1205° C WITH TBC	32.7
	ADV. COOLING DESIGN - 1205° C WITH TBC	32.4
	ADV. COOLING DESIGN - 1315° C WITH TBC	31.8

Figure 8. - Effect of thermal barrier addition and T_{IT} increase on cost of electricity (14).

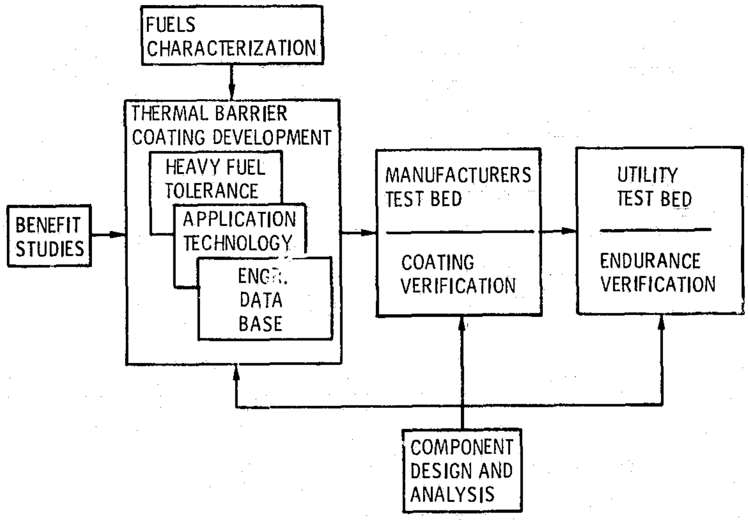

Figure 9. - Utility thermal barrier coating development logic.